

A DECADE OF AEROACOUSTIC RESEARCH AT NASA AMES RESEARCH CENTER

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SUMMARY

The rotorcraft aeroacoustic research accomplishments of the past decade at Ames Research Center are reviewed. These include an extensive sequence of flight, ground, and wind-tunnel tests that have utilized unique facilities to guide novel and pioneering theoretical research. Many of these experiments were of benchmark quality. They have been used to isolate the inadequacies of linear theory in high-speed impulsive noise research, have led to the development of new theoretical approaches, and have guided the emerging discipline of computational fluid dynamics to rotorcraft aeroacoustic problems. The results reported here have been achieved by a dedicated team of NASA researchers with, in many cases, the cooperation and assistance of personnel at the co-located Army rotorcraft center.

INTRODUCTION

Within its role of lead center for rotorcraft, NASA Ames Research Center (ARC) has had an active role in rotorcraft aeroacoustic research for the past 10 years. The major contributions have involved and drawn upon an extensive implicit knowledge of the integrated disciplines of the helicopter itself--including rotary-wing aerodynamics, performance, and dynamics. This knowledge, which resides at Ames because of its interdisciplinary rotorcraft research programs, has had a profound influence on rotorcraft acoustics. Significant advances have been made in theory development as well as in the utilization and development of unique acoustic testing techniques. In many cases, NASA researchers have worked with researchers of the Aeroflightdynamics Directorate, U.S. Army Research and Technology Activity, which is co-located at Ames Research Center. In effect, this close working arrangement substantially augmented the net effort in acoustic research--helping make it a very productive cooperative effort. The results presented here, were accomplished for the most part by NASA researchers. A companion paper addressing the Army contributions, with NASA support, is also presented at this conference.

Understanding, predicting, and reducing helicopter noise is important to both the civilian and military communities. For civilian operation, noisy helicopters limit the acceptability of their operation in many locations, thus limiting the commercial market for them. For military operations, loud detectable helicopters are easy to locate by even the most unsophisticated enemy. Early research into mechanisms that could produce the loud characteristic noises of helicopters yielded many hypotheses--even some remarkable theoretical approaches that attempted to

quantify different parts of the composite noise picture. Frequency-based methods were most often used to characterize rotor noise, dividing it into two general classes: harmonic and broadband noise sources (ref. 1). Little attention was paid to the phase of the resulting spectral analysis for either source of noise. In effect, noise levels were often compared with measured harmonics to decide if the mechanism being predicted matched the measured data.

Unfortunately, this frequency-domain approach (without phase information) did not lead to a complete understanding of the governing noise mechanisms. In hindsight, it turns out that many different sources of noise can combine to yield what appear to be acceptable theoretical-experimental comparisons of harmonic levels. In a great many cases, a more thorough comparison would have shown that theory and experiment did not really agree if the relative phases of the harmonic noise sources were included in the comparison. This was especially true for "impulsive noise sources"--unquestionably some of the most important sources of rotorcraft noise.

ACOUSTIC FLIGHT AND GROUND TESTING

In 1974, a new method of measuring helicopter noise was conceived (ref. 2) and implemented in a joint program with the Army Aeroflightdynamics Directorate, the Army Aviation Engineering Flight Activity, and ARC. A relatively quiet fixed-wing aircraft (OV-1C) was equipped with a tail-mounted microphone and an on-line data acquisition system to measure and record the in-flight external noise. Conditions known to produce helicopter noise were first established by the quiet fixed-wing aircraft, and the helicopter was then flown, at selected positions relative to it, in a station-keeping mode (fig. 1). Data gathered in this manner yielded the first undistorted picture of many helicopter noise sources. Using a simple plot of acoustic pressure time-histories it was possible to distinctly trace the various noise sources. The absence of ground reflections and Doppler shifts that are inherent in ground-measured data made the quantification of theoretical approaches possible.

This in-flight measurement technique was enhanced by the selection of the Lockheed YO-3A aircraft as the replacement for the OV-1. The YO-3A had been designed for use as a low-altitude observation aircraft during the Vietnam era and was, by design, an extremely quiet aircraft. The initial testing with a YO-3A, provided and supported by the Federal Bureau of Investigation (FBI), involved measuring the radiated noise of both the two UTTAS (YUH-60 and 61) and the two AAH (YAH-63, -64) prototypes during the Army's competitive evaluation. For these series of tests, the YO-3A was outfitted with three microphones and an in-flight data acquisition system. As a result of these early successes, in 1977 ARC acquired its own YO-3A acoustic research aircraft (fig. 2) and dedicated it to the task of measuring the acoustic emissions of low-speed flying vehicles--especially rotorcraft.

Over the past 8 years, ARC has operated this unique facility in a variety of acoustic research programs. These have included in-flight measurements of the new composite "747" main rotor blades for the AH-1 series helicopter, both tapered-tip

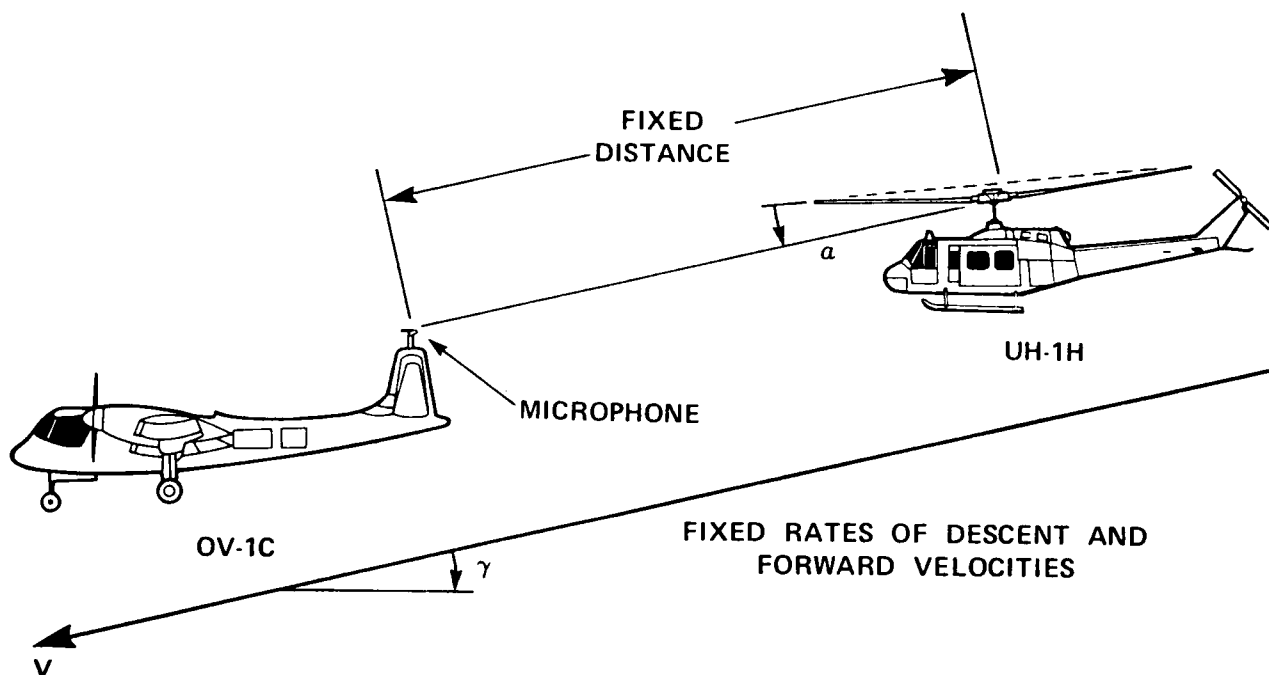


Figure 1.- Schematic of in-flight far-field measurement technique (from ref. 2).

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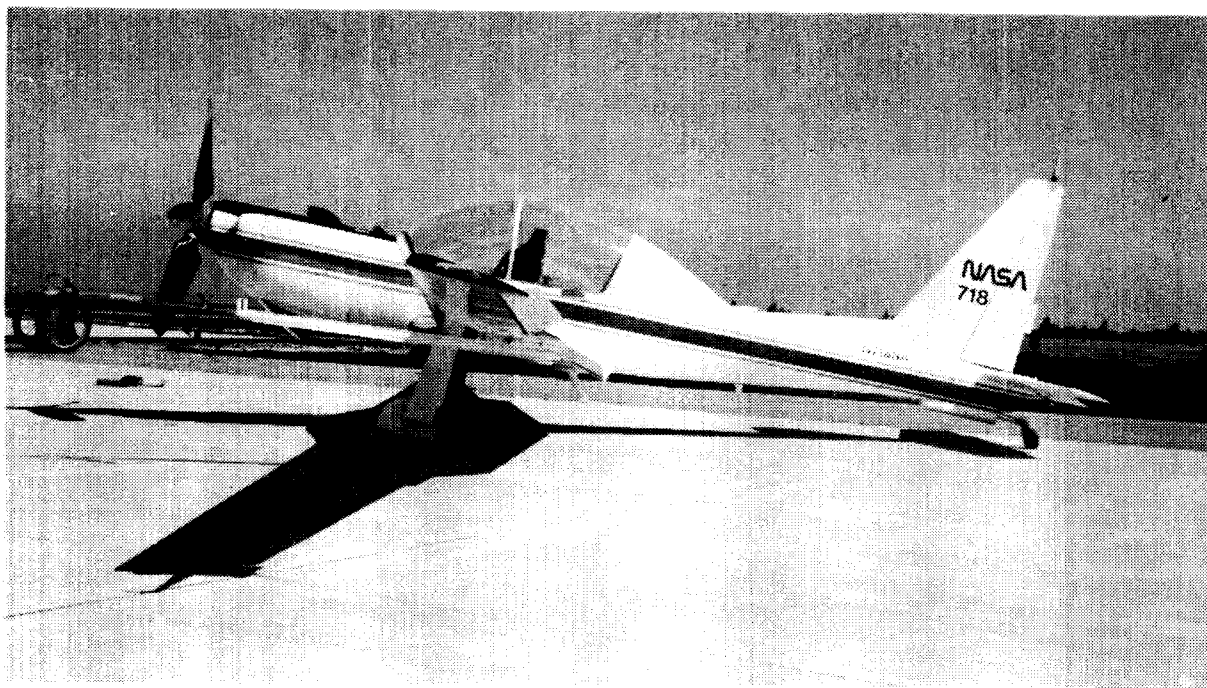


Figure 2.- Ames YO-3A quiet aircraft noise-measurement platform.

(fig. 3) and ogee-tip designs (refs. 3 and 4), 500-D four-bladed and two-bladed tail-rotor measurements, AH-1G pressure instrumented rotor blades (ref. 5), the AH-64 production baseline rotor, and the UH-60 production baseline rotor. In several of these programs, acoustic measurements were also made on ground arrays, using traditional flyover measurement techniques. During the 500-D and AH-1G tests, the flyover testing was done while flying in formation with the YO-3A, thereby completely documenting the emission characteristics of the subject helicopter. These two tests were conducted with assistance from NASA Langley Research Center (LRC), the lead center for acoustic research.



Figure 3.- YO-3A in-flight acoustic measurements of AH-1S helicopter with 747 taper-tipped fiberglass main-rotor blades.

The YO-3A Acoustic Research Aircraft will continue to be a vital facility for future aeroacoustic research programs. To enhance the capabilities of the YO-3A, measurement and station-keeping equipment are currently being upgraded. A helicopter acoustics laser positioning system (HALPS), is designed to give the precise position of the test aircraft relative to the YO-3A, and to measure the test aircraft's azimuth, elevation, and range to the YO-3A. The HALPS displays these values to the flight engineer aboard the YO-3A, and transmits them to the test aircraft, where a glide-slope indicator displays the information for the pilot, allowing him to establish and maintain precise formation flight. The flight-test engineer sets the desired aircraft formation with the use of a hand-held terminal. The accuracy of the system during bench testing has been of the order of ± 1 ft. Although HALPS has yet to be used to acquire flight data, it promises a significant improvement over the current optical ranging system. HALPS offers the additional benefit of having the range data recorded, along with the measured acoustic data, on the on-board tape recorder for use during postflight analysis.

The large number of different types of rotorcraft at Ames has facilitated many static and flyover noise-measurement programs, including the Rotor Systems Research Aircraft (RSRA) and XV-15 Tilt Rotor Research Aircraft. Acoustic measurements over microphone ground arrays and during tie-down tests led to both a near- and a far-field acoustic description of the XV-15 tilt-rotor aircraft (fig. 4). These near-field measurements were instrumental in demonstrating that cabin noise in the XV-15 cockpit was not a serious problem. Acoustic measurements taken at the Outdoor Aerodynamic Research Facility (OARF) have evaluated new higher-performance and lower-noise rotor blades for tilt-rotor aircraft (ref. 6). Some of these experimental findings have enhanced the knowledge base that has been used in the design of the Navy's V-22 Osprey--the world's first operational tilt-rotor aircraft.



Figure 4.- XV-15 tilt rotor tie-down noise tests.

WIND-TUNNEL ACOUSTIC MEASUREMENTS

The NASA Ames 40- by 80-Foot Wind Tunnel, or 40 x 80, has been the free world's largest low-speed wind tunnel. Before the 1980 upgrade, it had been used to gather important research data, including acoustic measurements on many full-scale rotorcraft systems. Because of the acoustically hard metal walls of the 40 x 80 test section, the acoustic measurements taken before the 1980 wind tunnel modification are not entirely representative of free-field data. The absolute sound levels are noticeably distorted by the reverberant characteristics of the test section. This distortion precludes detailed understanding of the acoustic field; however, comparison of overall noise levels is still possible. These early measurements were important because they were able to show the trends of sound levels with key parametric

changes. They were also used to assess gross noise level differences between new rotorcraft concepts.

One of the first attempts at measuring noise in the 40 x 80 was made in 1967 (ref. 7). A two-bladed Bell UH-1D helicopter was tested, in a standard and in a thin-tipped rotor blade configuration, at advancing-tip Mach numbers of 0.7 through Mach 1.0. Two microphones, which were placed in the test section about 1.1 and 1.4 diam upstream of the rotor, successfully recorded very loud impulsive noise levels at high advancing-tip Mach numbers. Although the absolute levels were contaminated by tunnel-reverberation effects, the noise level trends were basically correct, showing that impulsive noise harmonics increase dramatically with advancing-tip Mach numbers and that thin-tipped rotor blades can substantially reduce high-speed impulsive noise.

In the spring of 1977, the 44-ft-diam Sikorsky S-76 main rotor with four interchangeable tips (rectangular, swept, tapered, and swept-tapered) was tested in the 40 x 80 (refs. 8 and 9). This rotor system, shown in figure 5, mounted in the test section of the 40 x 80 incorporated state-of-the-art aerodynamics and represented Sikorsky Aircraft's design for the executive helicopter commuter market. The test envelope was extensive, spanning tunnel speeds of 30 to 175 knots, advance ratios of 0.075 to 0.4, advancing-tip Mach numbers of 0.64 to 0.97, rotor lift coefficients divided by solidity of 0.0 to 0.14, and rotor shaft angles of -10° to $+10^\circ$. Considerable data were acquired for simulated high-speed flight conditions of the S-76 rotor system. Above advancing-tip Mach numbers of 0.9 where transonic phenomena control the amplitude and waveform of the radiated impulsive noise, the swept-tapered (production) tip produced the lowest noise levels. These data showed that

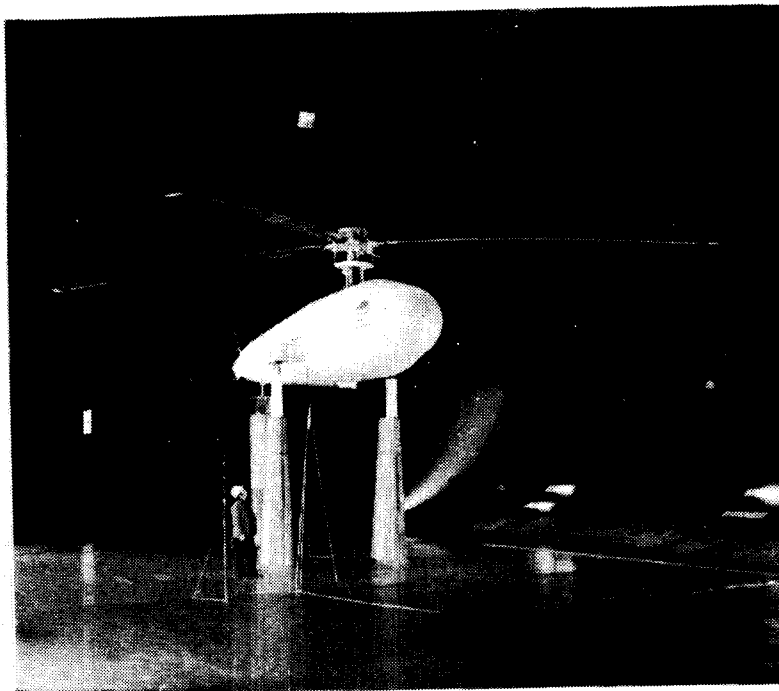


Figure 5.- S-76 main rotor mounted in 40 x 80 tunnel test section.

sweep and taper both help to reduce high-speed noise levels while at the same time improving the high-speed performance of the rotor.

Several entirely new rotorcraft configurations were also tested in the 40 x 80 for aerodynamics, dynamics, and noise. The XV-15 tilt-rotor research aircraft (forerunner of the multi-service V-22 Osprey aircraft) was tested with the rotors operating in a horizontal (helicopter mode) and in the vertical (airplane mode) plane. In the airplane mode (fig. 6) with the rotors operating at reduced cruise rpm, discrete harmonic noise levels were about 15 dB quieter than those measured in the helicopter mode (ref. 10). The higher rotational tip speeds and the unsteady aerodynamics of the rotors in the helicopter configuration were responsible for most of the increased noise levels.

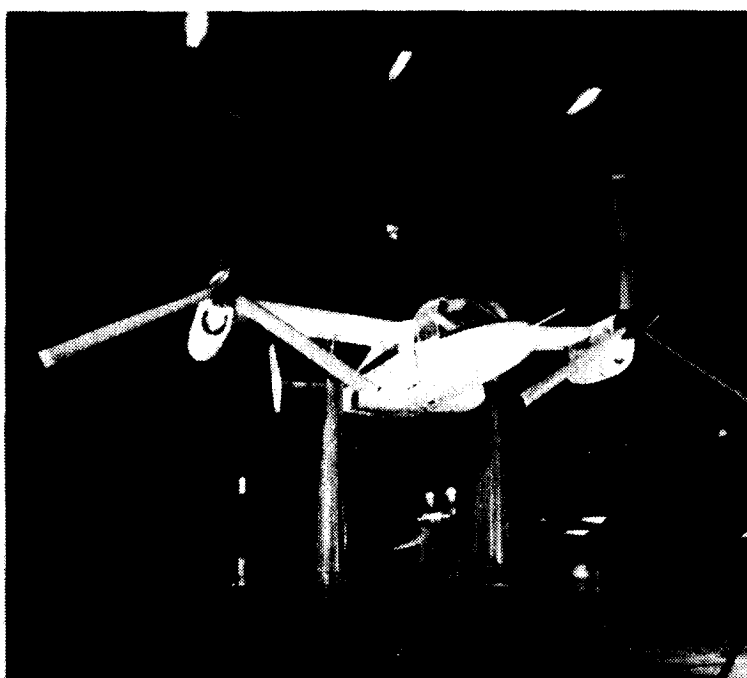


Figure 6.- XV-15 Tilt Rotor Research Aircraft mounted in the propeller mode in the 40 x 80 tunnel.

The use of "blowing" to control the lift of the main rotor was tested in two separate programs. In the summer of 1978, noise was measured on a 44-ft-diam four-bladed main rotor which used circulation control for collective and cyclic lift control (refs. 11-13). Mechanical controls also set the collective pitch of the blades; so collective lift depended on mechanical and pneumatic settings. Noise measurements in the 40 x 80, which were made from hover to 140 knots, showed that sound levels depended more on maximum jet-blowing velocity than on mechanical collective. Sound levels increased with increased blowing velocities. This increase was broadband in character, centered at about 2000 Hz, and believed to be a result of the blowing itself. At 60 knots the noise levels of this rotor were about 15 dB higher than those of a conventional rotor of similar performance, whereas at

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140 knots the levels were about 2 dB higher. Blowing was also used successfully in the spring of 1979 to control a 25-ft-diam circulation control, X-Wing rotor as it performed its first transition from helicopter to fixed-wing aircraft in a 40×80 tunnel test (fig. 7). The stoppable rotor had a circulation-control symmetrical airfoil with both leading- and trailing-edge slots to control cyclic and some of the collective lift. By carefully controlling the amount of blowing out of the slots, it was possible to transition the rotor to a stopped condition in forward flight while maintaining the lift and controlling moments on the rotor system. The measured sound levels of the rotor depended primarily on the jet-blowing velocity and secondarily on the forward speed. No characteristic impulsive noise was measured in this aircraft in the rotating (helicopter) configuration, although the test matrix included regions where blade-vortex interaction noise is present on conventional rotors. The sound levels, which were quite high at jet-blowing Mach numbers of about 0.7, decreased with either increased or decreased blowing.



Figure 7.- X-Wing rotor mounted in the 40×80 tunnel.

Just before the 40×80 tunnel was shut down for extensive modifications and upgrading in the summer of 1980, the advancing-blade concept (ABC) rotor was tested (ref. 14). The ABC demonstrator was a coaxial helicopter with counterrotating rotors that were 36 ft in diameter. The measured rotor sound levels were about 5 dB higher than for a conventional helicopter rotor (i.e., S-76 without a tail rotor) and in general exhibited more impulsive noise--probably because the rotors operate close to a more complex wake pattern. There is some indication that a part of this additional noise could be reduced by altering the lift distribution between rotor systems.

In mid-1980 an extensive upgrade of the 40 x 80 wind tunnel was begun which included re-powering the fan drive unit so the tunnel, in a closed-circuit operation, would reach a top speed of 300 knots and adding a new open-circuit 80- by 120-ft leg for large-scale low-speed testing (top speed approaching 100 knots). A lower maximum drive fan tip speed of 377 ft/sec plus a 6-in. fiberglass acoustic lining for the closed circuit test section were two major acoustic improvements. The 6-in. acoustic liner (fig. 8) was mounted behind a perforated plate to give near-anechoic (without echoes) conditions above 500 Hz in the test section. There was an unfortunate structural failure of the tunnel in 1982, which will delay operational testing until the spring of this year. Preliminary calibration measurements of the flow quality and background-noise levels to date are encouraging (as of February 1987 the tunnel has attained a top speed of 295 knots) and do ensure the 40 x 80 an important role in the rotorcraft acoustic testing world. The 80 x 120 leg of the 40 x 80 also may be useful for acoustic testing. Its enormous test section will be treated with 6 in. of acoustic material on the floor and ceiling and 10 in. on the sidewalls to help reduce the noise radiated to the surrounding community. A significant benefit of this treatment is an improvement in the test section acoustical properties, making it possible to make far-field noise measurements of model-scale and some full-scale rotor systems in the test section.

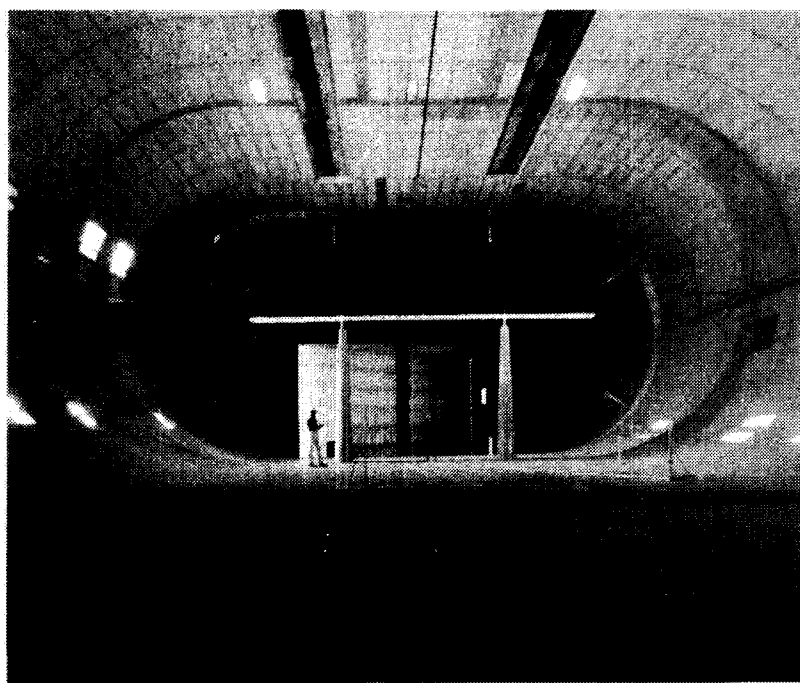


Figure 8.- 40- by 80-ft test section with acoustical liner.

Studies are also under way to investigate whether the size of the 80 x 120 and the low fan-drive noise can be used in a further modification of the 80 x 120 test section to make it the world's largest anechoic wind tunnel (ref. 15). The inherently large size of this modified facility would allow near-anechoic properties at low frequencies and large distances--factors that are important in source-noise reduction for acoustic detection of rotorcraft.

ROTORCRAFT SCALING

Although prediction of a rotorcraft's noise levels from a first-principle approach is the ultimate goal of much of the acoustic research done in the United States, a more immediate practical goal is the use of small-scale model-rotor testing to assess the effect of design changes on the radiated noise. Scale-model testing offers the advantages of easier tunnel access (because of the availability of more wind tunnels which can test small-scale models), and a more anechoic environment (because the scaling raises the frequencies of the rotorcraft noise sources allowing them to be more easily absorbed by acoustic wall treatment). However, it first must be conclusively demonstrated that the model-scale results accurately represent the full-scale acoustic phenomena of interest. To this end, much of the work in industry and at ARC has been to try to quantify how well full-scale acoustic phenomena can be duplicated in model scale. Unfortunately, no really clear general picture has emerged, although the researchers who are involved are still optimistic.

Over the years, small-scale models of rotor systems have been used to assess differences in performance and noise of competing configurations (refs. 16-19). The best results have been obtained when the model test has been run to investigate the effects of geometric changes on a previously isolated noise source, that is, high-speed impulsive noise or blade-vortex interaction noise. It is hoped that the differences in noise levels that were measured in model scale are similar to those that would have been measured on the full-scale rotorcraft. Unfortunately, many of the documented tests showing that these model-scale results do represent full-scale noise levels are not very convincing. If the data were taken in non-anechoic measurement halls, it must be corrected for reverberation effects. Adjustments are made to the measured sound-pressure levels in frequency bands (usually 1/3 octaves) resulting in levels that are corrected to free-field data. The full-scale data, on the other hand, are usually corrected for ground or wall reflections, and wind and temperature effects. It is not unusual to have differences between model and full-scale data in these adjusted levels of about 6 dB (ref. 19). This is not an adequate verification that the rotor noise sources have been faithfully reproduced in model scale. Differences of 6 dB translate into possible aerodynamic source-level differences of 100%.

More definitive scale-model rotor testing is under way. Research performed under contract to Ames by the United Technologies (ref. 20), as well as scale-model testing by Ames researchers (ref. 21), has yielded mixed results. The 1/6-scale model S-76 rotor (fig. 9) did not faithfully match full-scale data. In conditions simulating forward-flight descents typical of a landing approach, sound-pressure differences of up to 10 dB were shown between model- and full-scale wind tunnel data. The 1/20-scale models of reference 20 have shown even poorer correlation with full-scale data.

It is tempting to conclude that model-scale tests do not reproduce full-scale phenomena, but other researchers have shown more promising results (refs. 22 and 23). Under certain operating conditions, high-speed impulsive and blade-vortex

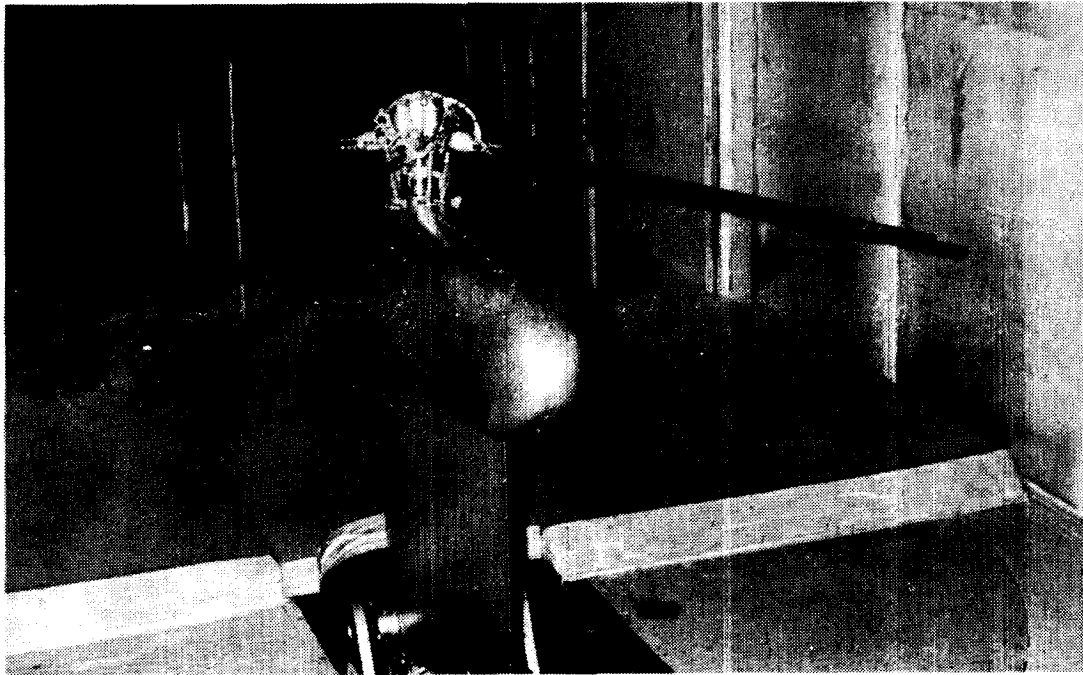


Figure 9.- One-sixth-scale S-76 model rotor mounted in Ames acoustically treated 7- by 10-ft wind tunnel.

interaction noise were reproduced quite faithfully in model scale. Why then are there such mixed results?

Part of the answer may lie in the extreme difficulty of doing carefully controlled model-scale and full-scale acoustic testing. Aside from the corrections for the non-ideal acoustic environment, the operating conditions known to produce much of the terminal-area approach noise, which is due to blade-vortex interaction, are quite sensitive to the separation distances between the rotor blades and the closely passing tip vortices. Thus, it is very important that all model-scale and full-scale trim conditions be matched exactly. The most difficult condition to match is the actual tip-path-plane angle measured with respect to the local velocity vector. This angle is influenced by wind-tunnel wall corrections--something not accounted for in several of the less successful tests reported here. It is probable that a better match between model- and full-scale data is possible. New testing comparing scaled rotors under more controlled and definitive conditions in environments that do not substantially alter the measured sound-pressure levels is currently under way.

THEORETICAL PREDICTIONS

An important goal of the acoustic research efforts of the past 10 years has been to more accurately predict, from first principles, helicopter noise. Ames has

played a key role in this effort by demonstrating the strengths and weaknesses of current theoretical efforts. This has, in many cases, led to new theoretical approaches or methods for predicting the radiated noise.

For several reasons, large, comprehensive, multidisciplinary mathematical models for predicting far-field rotorcraft noise have not been emphasized at Ames. First, very simple controlled experiments of isolated noise sources (refs. 24 and 25) have shown that existing linear theory does not predict the radiated noise to within ± 3 -6 dB. One cannot expect the accuracy of a comprehensive mathematical model to be any better than any of its components. Secondly, the noise radiation field is very sensitive to the aerodynamic flow field surrounding the blade, which is not known to the degree necessary to predict the global acoustic field. The comprehensive aerodynamic-dynamic models that are considered to be state-of-the-art (e.g., C-81, CAMRAD) are known to be inadequate for noise-prediction purposes by themselves. They cannot predict the higher harmonic air-load flow-field environment of the operational helicopter. The aeroacoustic researchers at Ames have chosen a different approach. Emphasis has been placed on developing those theories or models that isolate key aerodynamic and acoustic phenomena that can be verified with simple, straightforward model-scale or full-scale experiments, in flight, in static tests, or in wind tunnels.

One of the first efforts in this regard was that of Arndt and Borgman (ref. 26). Using measured acoustic data taken in the 40×80 tunnel of a rotor operating at high tip speed, they showed that the radiated noise could be attributed to the increasing drag rise caused by shock waves on the advancing side of the rotor disk. Because only sound-pressure levels were directly compared in the theory-experiment, the results did look encouraging. If actual time-histories had been compared under these conditions, the comparison would not have been as favorable. This was verified in some of the in-flight measurements that were flown a few years later and discussed earlier in this paper. Nevertheless, these early researchers showed that noise levels grew tremendously as the rotor's advancing-tip Mach number approached 1--an experimental observation noted in the early 40×80 acoustic testing. However, their theory never could explain why the high-speed impulsive noise levels were not sensitive to thrust variations, a key finding of these earlier experiments.

Another attempt to predict rotorcraft noise was made in 1978 (ref. 27). A simple theoretical linear-thickness monopole model was developed that was used to generate waveforms of the Sikorsky S-76 rotor system in high-speed flight. These calculations were compared with averaged time-domain S-76 data taken in the 40×80 tunnel. Comparison with experiment was generally encouraging because the pulse shapes looked similar to the measured data. However, quantitative comparisons were difficult because of the reverberant environment in the untreated test section in these early tests. Nevertheless, the importance of blade-thickness effects on the radiated noise was clearly shown by these early computations.

The usefulness of using linear acoustic models to describe the radiated acoustic field of helicopter rotors in high-speed flight was called into question by some fundamental high-tip-speed model-scale data (ref. 24). Large discrepancies in level

and pulse shape between theory and experiment were shown by using data from these very controlled fundamental experiments. The discrepancies were tied to transonic aerodynamic effects which are represented as quadrupole source terms in the acoustic analogy formulation (ref. 28). Including these nonlinear terms improved the correlation with experiment in these benchmark hover computations and showed that predicting high-speed impulsive noise was impossible unless the nonlinear effects of the noise generation and radiation process were modeled (ref. 19). Therefore, the acoustic analogy approach was not vigorously pursued for this source of impulsive noise. Instead, computational fluid dynamic (CFD) codes were adopted and expanded to address the high-speed acoustic problem, as well as the transonic aerodynamic problem. In essence, acoustics and aerodynamics were directly linked, implying that further improvements in high-speed far-field noise prediction were dependent on the accurate prediction of the local transonic flow over the blade.

Another significant theoretical development was recently made by Mosher (refs. 30-32). Mosher developed a new, very general method to examine the effects of wind-tunnel walls on discrete frequency noise, such as the low-frequency harmonic noise typical of rotorcraft. The theoretical model consists of an arbitrary, known harmonic acoustic source of finite dimension inside an infinite duct of constant cross-sectional area (representing the tunnel test section) with uniform subsonic flow (see fig. 10). An impedance boundary condition on the duct wall allows the wall to absorb some of the incident acoustic energy. Numerical solutions are found by matching an acoustic panel method in a control volume around the helicopter rotor to a modal series representation of sound in the duct far from the rotor. Because this scheme allows arbitrary duct shapes and arbitrary sources, sound in a complicated system may be analyzed.

Some representative results from this new theory are shown in figure 11 for a horizontal plane that is 0.4 of a radius below the rotor. Contour levels of the fundamental rotor harmonic peak amplitudes are modified by the test section of the wind tunnel at these very low frequencies. Very close to the rotor, the sound field

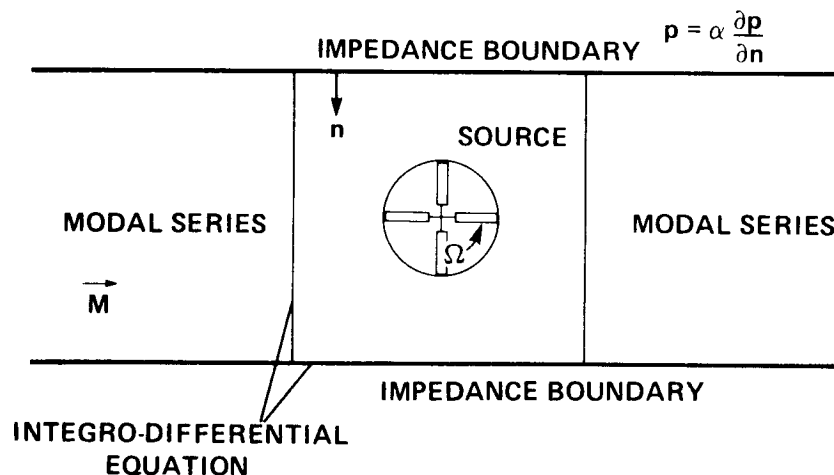


Figure 10.- Computational model of a rotor in a wind tunnel.

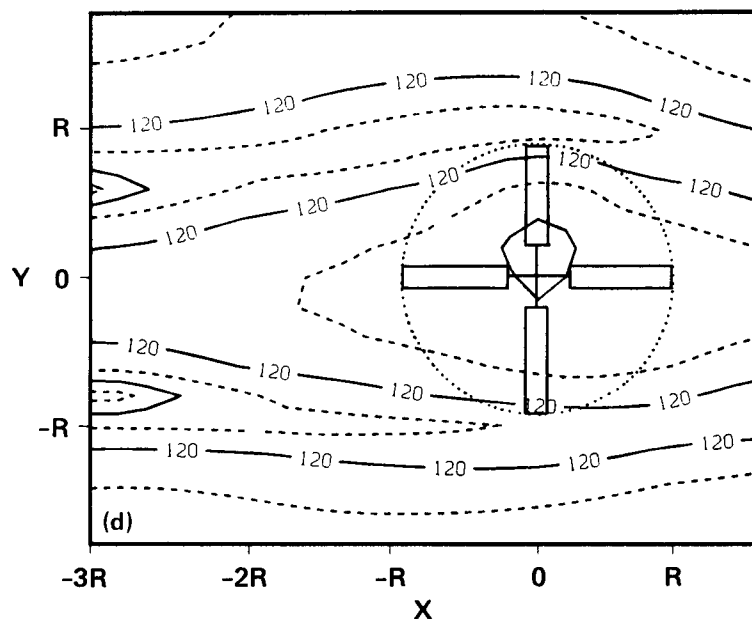


Figure 11.- Representative results: contours of sound-pressure levels in a horizontal plane 0.4 R below the rotor plane.

in the wind tunnel resembles the sound field in unbounded space. However, at distances required for mid- to far-field noise measurements, the sound field has characteristics that are predominantly determined by modal propagation in the duct. Wind-tunnel test-section wall acoustic characteristics and the relative size of the rotor disk in relation to the test-section dimensions determine how far from the rotor useful acoustic measurements can be made that closely match free-field conditions. The reverberation problem is most severe at the lower blade passage frequencies where it is very difficult to have good tunnel-wall absorption.

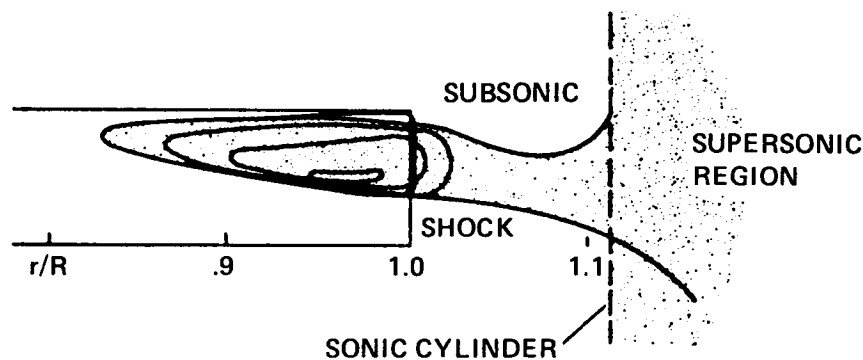
The implications of this work are significant. Once the theoretical model has been verified experimentally, it can be used to pick measurement positions in existing wind tunnels to minimize duct wall effects, to determine how to acoustically treat existing wind tunnels to measure rotorcraft noise in a specific frequency range, and to help modify existing wind tunnels to enhance their measurement properties. The method is computationally efficient at low frequencies--where the reverberation problem is most severe for all existing wind tunnels.

COMPUTATIONAL FLUID DYNAMICS

Ames has pioneered the application of computational fluid dynamics (CFD) to helicopter aerodynamic and acoustic problems. The close connection between aerodynamics and acoustics as they relate to impulsive noise was first made in some benchmark hover experiments performed by the Army in 1978 (ref. 28). A model-scale UH-1H rotor was tested in an anechoic hover chamber while far-field acoustics and local

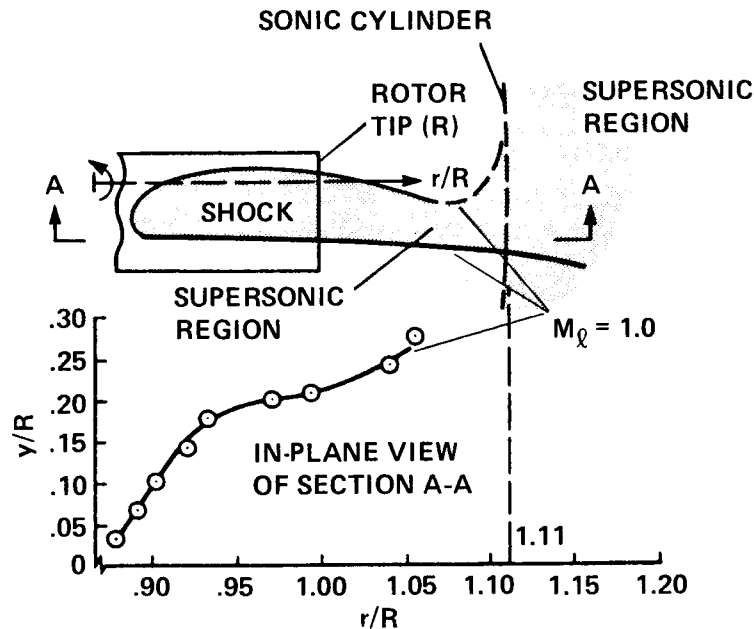
blade-flow measurements were made simultaneously. At hover tip Mach numbers approaching 0.9, local shock waves, which first appeared at lower subsonic tip Mach numbers at about 90% radius, "delocalized" to the acoustic far-field. In effect, localized nonlinear aerodynamic effects were directly propagated to the acoustic far-field--helping explain why linear acoustic analogies could not adequately predict the rotor's acoustic radiation. It was shown (ref. 29) that the criterion for acoustic "delocalization" was the local flow Mach number, as viewed by an observer in a coordinate system rotating with the blade, must be equal to or greater than 1 in the continuous region from just in-board of the rotor tip to the microphone located in the acoustic far-field.

The Army and ARC already had significant CFD efforts under way to predict the potential flow field of high-speed rotorcraft (refs. 33 and 34). These efforts were modified to predict the entire flow field in the vicinity of the rotor to estimate regions where delocalization might occur. A first attempt at predicting this phenomenon was reported in reference 35 and is shown in figure 12(a). Experimental measurements of the same rotor (fig. 12(b)) show that these CFD methods predict the transonic acoustic field quite well (ref. 28). These particular computations were done by Bell Helicopter using the NASA Ames ROT-22 full-potential quasi-steady code. This same code has been used extensively at Bell, with NASA assistance, to help in the design of new tip shapes to avoid the loud impulsive noise associated with the delocalization phenomenon. In the past 4 years, new full-potential and Euler finite difference codes have been integrated with integral rotor-wake models to predict the full, unsteady, high-speed aerodynamic and acoustic near-fields (refs. 36-38). A comparison of the full-potential code results with local blade-surface pressures for a nonlifting rotor operation at high advance ratios and at high advancing-tip Mach numbers is shown in figure 13 (ref. 36). The good agreement between the unsteady computations and experimental pressure distribution measurements are typical of both full-potential and Euler codes for the nonlifting rotor. Shock positions and strength are predicted quite well on the rotor's advancing side. The importance of using unsteady computational results can also be seen in the same figure. The quasi-steady results (dotted curves) do not predict shock locations well at angles other than 90°.



(a) Theory.

Figure 12.- Theory-experiment comparison of delocalization of shock waves to the acoustic far field of a hovering rotor.



(b) Experiment

Figure 12.- Concluded.

A similar comparison for the lifting rotor is shown in figure 14 for the high advancing-tip Mach numbers at high advance ratios. In this case, the rotor's trailed tip-vortex system is modeled in the computational procedure. At these high advance ratios, this wake system is fairly far from the advancing rotor. Under this condition, agreement between the full-potential (ref. 37) and experiment (ref. 39) is generally good, as shown. However, at lower advance ratios when the wake system is close to the advancing blade, good agreement between computation and experiment is less likely. Additional work is under way to more accurately model the rotor-wake system in these cases.

It should be noted that because the flow field is highly nonlinear, there is virtually no method of predicting the flow field of a high-speed rotor other than the CFD approach. The introduction of CFD methods to this problem by Ames researchers has provided the helicopter industry with powerful new tools to help design high-speed rotorcraft that are efficient and quiet. An example of the use of CFD in this regard is the aerodynamic and acoustic design of the main rotor blades for the Bell OH-58 advanced helicopter improvement program. Bell engineers, using the ARC developed ROT-22 full-potential code (ref. 34), designed a new OH-58 main rotor that yielded increased high-speed performance and less noise. Local blade section and planform characteristics near the tip of the advancing rotor were carefully chosen to avoid "delocalization," and hence avoid impulsive noise radiation in high-speed flight.

The impulsive noise caused by blade-vortex interaction (BVI) has also been seriously addressed by NASA and the Army using CFD techniques over the past few

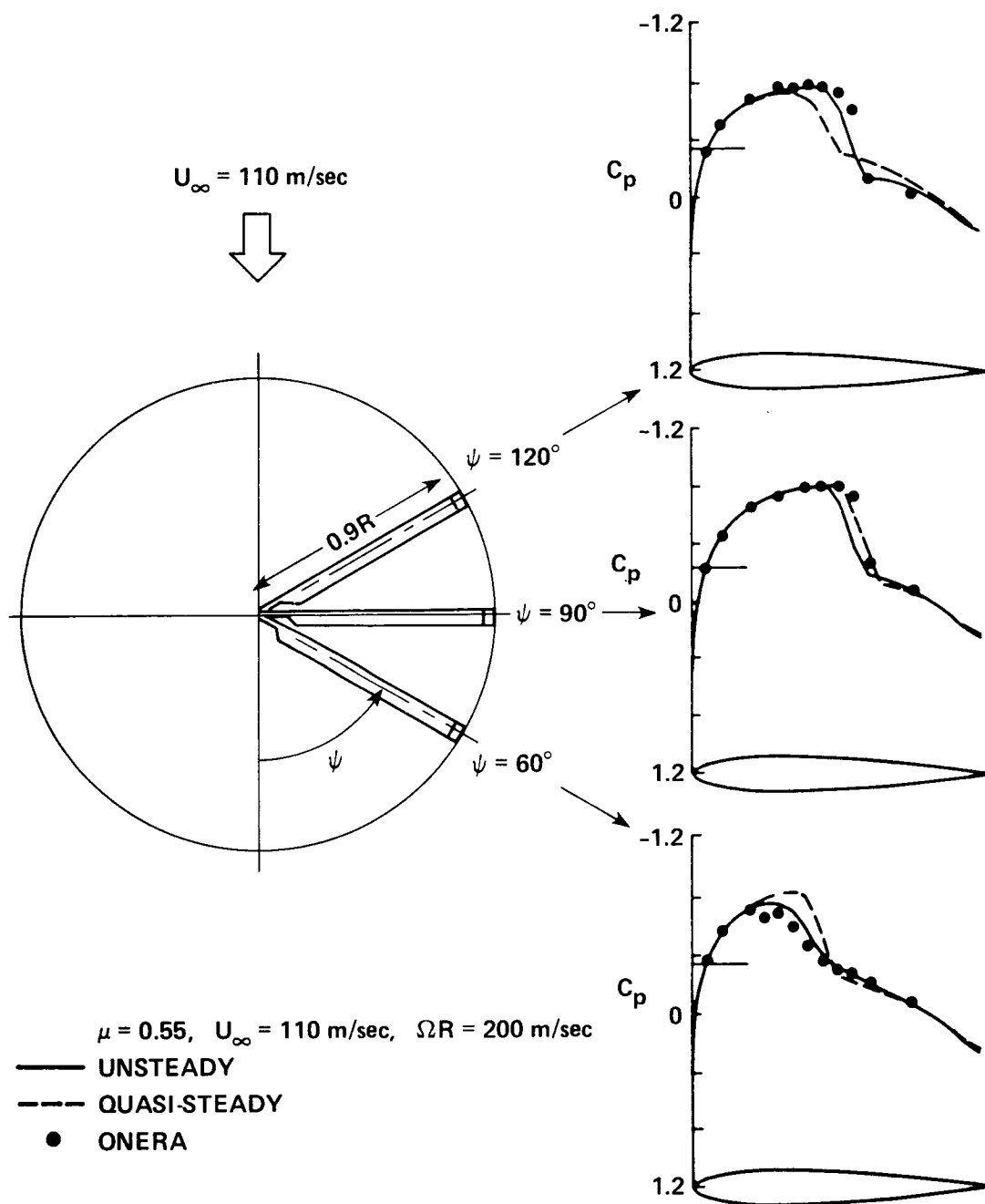


Figure 13.- Comparison of full potential CFD computations with experimental measurements for a high-speed nonlifting rotor in forward flight.

years. Experimental findings have suggested that part of the problem can be represented in two dimensions by considering a two-dimensional blade section interacting with a two-dimensional vortex. The first studies, which utilized an ARC-developed small-perturbation potential code with coarse grids (ref. 40), clearly showed the aerodynamic details of the interaction process near the two-dimensional blade section. However, the acoustic waves that radiated away from the airfoil were weak and

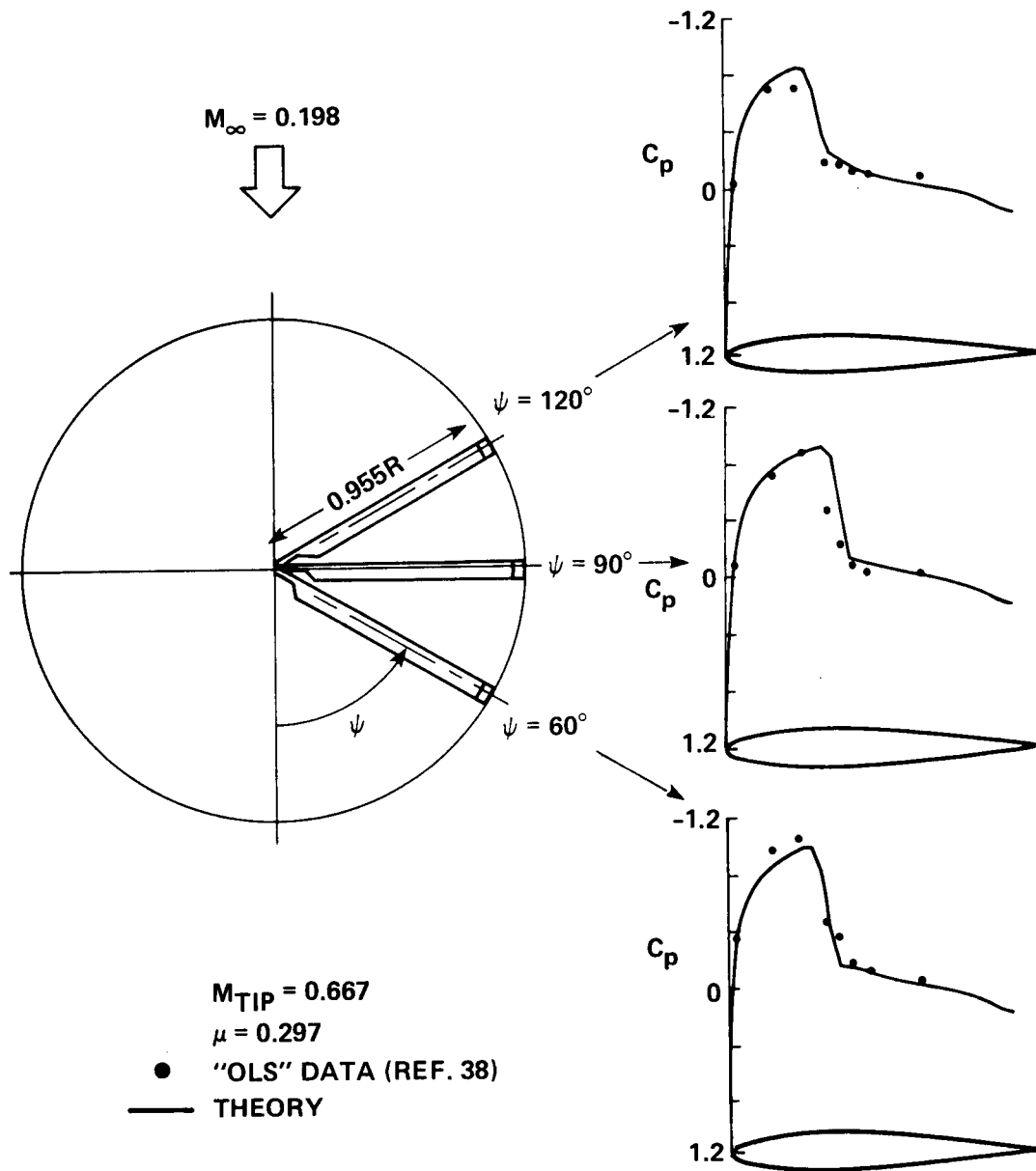


Figure 14.- Comparison of full potential CFD computations with experimental measurements for a high-speed lifting rotor.

were subject to grid-smearing by the CFD process itself. Nevertheless, it was possible to trace acoustic waves radiating to the boundaries of the computation field. Several authors have since pursued this two-dimensional CFD problem with more powerful CFD codes and cleverer ways of looking for radiating acoustic waves (ref. 41). These codes have also been installed on industry computers and used to help design airfoils that reduce the shock-like disturbances that can occur near the leading edge of the airfoil during BVI. Sikorsky Aircraft, using an ARC-developed code (LTRAN2-HI), developed the 1095 RN airfoil to help reduce the impulsive noise that is radiated during blade-vortex interaction (BVI). The airfoil has a specially

designed leading-edge contour that according to CFD computations, reduces the local shock waves during BVI. However, there is still some uncertainty about the effectiveness of these leading-edge design changes in reducing the radiated far-field noise. New theoretical computations and experiments are planned to help ascertain whether these near-field acoustic results are indicative of what happens in the acoustic far-field.

An extremely interesting and complete representation of the two-dimensional BVI problem was recently given by Rai (ref. 42). Using a very high-order CFD scheme on the Cray X-MP computer, he was able to capture the unfolding acoustic radiation problem in great detail (fig. 15). His method used the Navier-Stokes equations and did not describe, a priori, the trajectory or the unfolding structure of the vortex. Instead, the entire unsteady transonic BVI problem was simulated and solved in a very accurate manner. As computers become faster and larger in the next few years, it will be possible to design airfoils and hence complete rotor systems to reduce the noise radiated by rotorcraft.

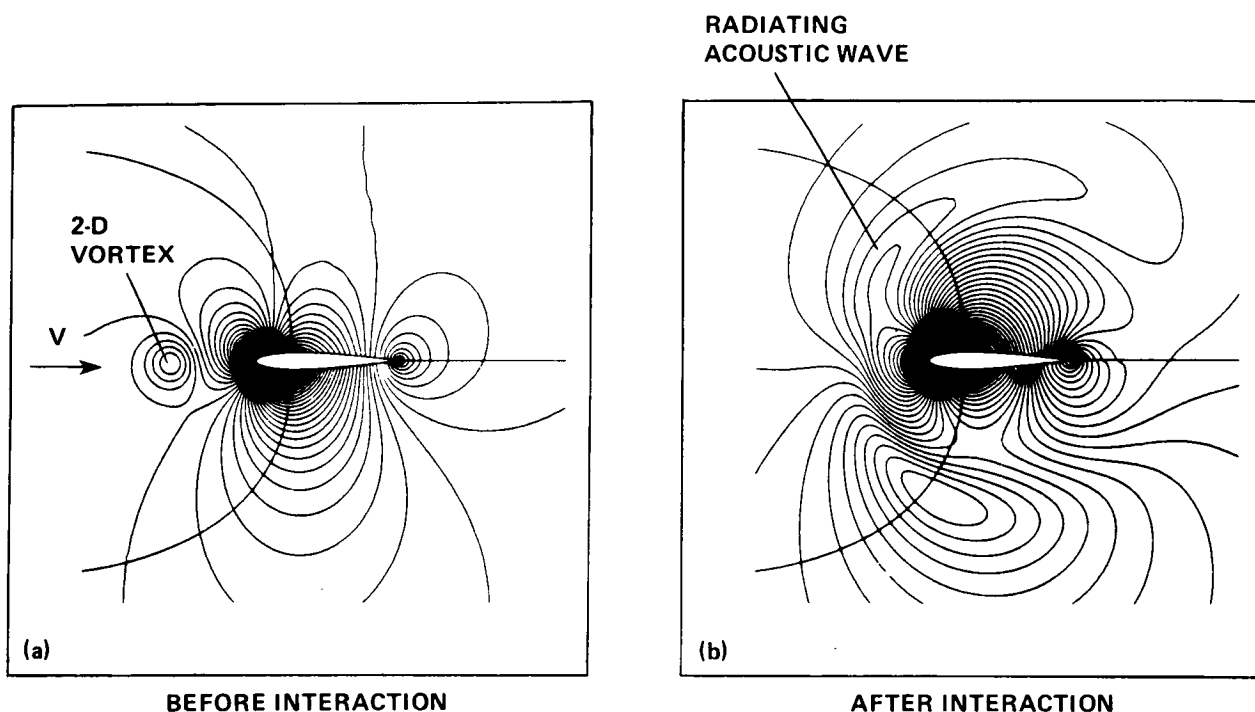


Figure 15.- An acoustic wave radiating from a 2-D airfoil during blade-vortex interaction.

CONCLUDING REMARKS

This paper has reviewed the important rotorcraft aeroacoustic advances that have been made at Ames Research Center over the past decade. On a fundamental research level, there have been some key experimental, as well as theoretical breakthroughs. It is very difficult to clearly separate the advances into two distinct

classes because theory and experiment have been integrated in most rotorcraft programs. Nevertheless, it's probably fair to say that the findings of a few key experiments have helped change the understanding of the important rotorcraft noise-generating mechanisms and have led to the development of new and fruitful theoretical approaches. In particular, the following accomplishments stand out.

1. The development of the YO-3A quiet aircraft as an in-flight acoustic measurement platform. It has been used to measure the radiation characteristics of many research and prototype aircraft.
2. The development of the National Full-Scale Aerodynamics Complex (NFAC) (40 x 80 x 120), the world's largest aeroacoustic testing facility. The low fan-drive tip speed, the approximately 300-knots forward-speed capability, and the extremely large test sections make the complex unique for rotorcraft aeroacoustic research at model- and full-scale.
3. A new theoretical method for predicting the effect of wind-tunnel walls on the radiation patterns of low-frequency rotorcraft noise. The method will be used to help design appropriate testing techniques and tunnel treatment for low-frequency noise--the most important contributor to rotorcraft signature reduction efforts.
4. The discovery of the causes and the mechanisms of impulsive noise radiation. Through extensive wind-tunnel and flight testing, improved theoretical approaches were developed and verified that more accurately described the impulsive noise phenomena. An important finding of the improved prediction techniques was that it was necessary to compare amplitude and phase (or compare time-histories) of the measured and predicted acoustic radiation.
5. The application of computational fluid dynamics (CFD) to rotorcraft acoustic problems. A new CFD capability has been developed and transferred to United States industry to help solve high-speed and blade-vortex interaction impulsive noise problems.

Besides providing the tools, techniques, and facilities, Ames Research Center has been working with industry to try to reduce rotorcraft noise through design. One notable example of these efforts is the design of the rotor blades for the Bell OH-58 advanced helicopter improvement program. Knowledge of the delocalization mechanism, together with a direct application of CFD to the high-speed acoustic problem, led to rotor tips that were designed for increased performance and reduced noise. Another example of the application of this new technology is the design of the SC-1095 RN airfoil. Working with a CFD code that was developed at Ames, and using the knowledge acquired in fundamental research programs, Sikorsky Aircraft engineers have developed an airfoil section that promises to reduce rotorcraft blade-vortex interaction noise. Experimental efforts are now planned to verify the theory in a full-scale aeroacoustic test in the 40 x 80 tunnel.

Many of these results have been transferred to industry through the normal publication process and through cooperative and contracted programs with other government agencies and industry. Of particular significance in this regard is the

excellent cooperative research efforts that have existed between the U.S. Army Aeroflightdynamics Directorate and ARC. In some of the Ames research accomplishments it is very difficult to sort out the contributions of the Army and Ames scientists. Also of note is the positive role that the NASA/AHS/Industry National Rotorcraft Noise Reduction program has had on the level of support for rotorcraft aeroacoustic research within the government and United States industry. This program has funded the engineers who are responsible for applying these advances to the design and operational problems of the rotorcraft industry, thereby expediting the technology transfer from the government research laboratories to industry.

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